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# RESEARCH MEMORANDUM

## AERODYNAMIC CHARACTERISTICS OF TAPERED WINGS HAVING ASPECT RATIOS OF 4, 6, AND 8, QUARTER-CHORD LINES SWEPT BACK 45°, AND NACA $63_1$A012 AIRFOIL SECTIONS

TRANSONIC-BUMP METHOD

By Edward C. Polhamus and Thomas J. King, Jr.

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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## SUMMARY

As part of a transonic research program conducted by the National Advisory Committee for Aeronautics, a series of wings is being investigated in the Langley high-speed 7- by 10-foot tunnel over a Mach number range of 0.70 to 1.15 by use of the transonic-bump technique.

This paper presents lift, drag, pitching-moment, and root-bending-moment data of wings having aspect ratios of 8, 6, and 4, quarter-chord lines swept back 45°, and NACA $63_1$A012 airfoil sections parallel to the plane of symmetry. The wings having aspect ratios of 6 and 4 were obtained by removing portions of the tip of the wing having an aspect ratio of 8 and, therefore, the taper ratios were 0.45, 0.56, and 0.68.

All three wings, and especially the wing having an aspect ratio of 8, were characterized at low lifts by large reductions in lift-curve slope and very large forward movements of the aerodynamic center in the Mach number range from about 0.85 to about 1.0. However, above a lift coefficient of about 0.10 the pitching-moment characteristics as a function of Mach number were considerably better especially for the wing having an aspect ratio of 4. All three wings had a rather gradual drag rise starting in the Mach number range from about 0.90 to about 0.95.

## INTRODUCTION

A series of wings and wing-fuselage combinations is being investigated in the Langley high-speed 7- by 10-foot tunnel to study the effects of wing geometry on longitudinal stability characteristics at transonic speeds. A Mach number range of 0.70 to 1.15 was obtained by use of the transonic-bump technique.

This paper presents the results of an investigation of force and moment characteristics of wings having aspect ratios of 8, 6, and 4, quarter-chord lines swept back 45°, and NACA $63_1$A012 airfoil sections parallel to the plane of symmetry. The wings having aspect ratios of 6 and 4 were obtained by removing portions of the tip of the wing having an aspect ratio of 8 and, therefore, the taper ratios were 0.45, 0.56, and 0.68.

Although the Reynolds numbers of the tests were extremely low (about 570,000) and the spanwise Mach number gradient was rather large, it is felt that the results will give at least a qualitative indication of the difficulties that may be anticipated with relatively thick wings in the transonic speed range.

## COEFFICIENTS AND SYMBOLS

$C_L$ lift coefficient $\left(\frac{\text{Twice semispan lift}}{qS}\right)$

$C_D$ drag coefficient $\left(\frac{\text{Twice semispan drag}}{qS}\right)$

$C_m$ pitching-moment coefficient referred to $0.25\bar{c}$ $\left(\frac{\text{Twice semispan pitching moment}}{qS\bar{c}}\right)$

$C_B$ bending-moment coefficient at plane of symmetry $\left(\frac{\text{Root bending moment}}{q\frac{S}{2}\frac{b}{2}}\right)$

$q$ effective dynamic pressure over span of model, pounds per square foot $\left(\rho V^2/2\right)$

$S$ twice wing area of semispan model, square feet

| | |
|---|---|
| $\bar{c}$ | mean aerodynamic chord of wing, feet, based on relationship (using the theoretical tip) $\left(\frac{2}{S}\int_0^{b/2} c^2 dy\right)$ |
| A | aspect ratio $\left(\frac{b^2}{S}\right)$ |
| c | local wing chord, feet |
| b | twice span of semispan model, feet |
| y | spanwise distance from plane of symmetry |
| $\rho$ | air density, slugs per cubic foot |
| V | free-stream velocity, feet per second |
| M | effective Mach number over span of model |
| $M_l$ | local Mach number |
| $M_a$ | average local Mach number, chordwise |
| R | Reynolds number of wing based on $\bar{c}$ |
| X | distance along airfoil chord, percent chord |
| Y | airfoil ordinate, percent chord |
| $\alpha$ | angle of attack, degrees |
| $C_{D_{C_L=0}}$ | drag coefficient at zero lift |

## MODELS

The basic model had $45^\circ$ of sweepback referred to the quarter-chord line, an aspect ratio of 8.0, a taper ratio of 0.45, and an NACA $63_1$A012 airfoil section parallel to the plane of symmetry. The models obtained by removing portions of the tip of the basic wing had aspect ratios of 6.0 and 4.0 with taper ratios of approximately 0.56 and 0.68, respectively. All three wings had tips of revolution. Details of the models which were constructed of beryllium copper are presented in figure 1 and the airfoil ordinates (obtained from reference 1) are given in table I. The end plate shown in figure 1 was used to minimize leakage effects due to the angle-of-attack cutout.

## APPARATUS AND TESTS

The tests were conducted in the Langley high-speed 7- by 10-foot tunnel, an adaptation of the NACA wing-flow technique being used to obtain transonic speeds. The method used involves the mounting of a model in the high-velocity flow field generated over the curved surface of a bump located on the tunnel floor. (See reference 2). The model is mounted on an electrical strain-gage balance enclosed in the bump. The lift, drag, pitching moment, and bending moment are measured with potentiometers. A photograph of the model mounted on the bump is shown as figure 2.

Typical contours of local Mach numbers in the region of the model location on the bump, obtained from surveys with no model in position, are shown in figure 3. There is a Mach number gradient which results in differences of 0.06 to 0.08 over the span of the semispan model of aspect ratio 8 at the low Mach numbers and of 0.10 to 0.11 at the highest Mach numbers. The chordwise Mach number gradient is generally less than 0.01. The long-dash lines shown near the wing root represent a local Mach number 5 percent below the maximum value and indicate the extent of the bump boundary layer. The effective test Mach number was obtained from contour charts similar to those presented in figure 3 from the relationship

$$M = \frac{2}{S}\int_0^{b/2} cM_a dy$$

The variation of test Reynolds number with Mach number is shown in figure 4.

Force and moment data were obtained through a Mach number range of 0.70 to 1.15 and an angle-of-attack range of $-2^{\circ}$ to $10^{\circ}$ except for the wing having an aspect ratio of 4 where the maximum angle was $6^{\circ}$.

The end-plate tare corrections to the drag were obtained through the test Mach number range at an angle of attack of $0^{\circ}$ by testing the model configuration without an end plate. A gap of about 1/16 inch was maintained between the wing root and the bump surface, and a sponge-wiper seal (fig. 5) was fastened to the wing butt beneath the surface of the bump to minimize leakage. End-plate tares obtained in previous investigations were found to be constant with angle of attack, and the tares obtained at zero angle of attack in the present investigation were applied to all drag data. No end-plate-tare corrections were applied to the lift, pitching moment, and bending moment. Jet-boundary corrections have not been evaluated since the boundary conditions to be satisfied are not rigorously defined. However, inasmuch as the effective flow field is large compared with the span and chord of the model, these corrections are believed to be small. The basic data have not been corrected for deflection under load; however, the lift-curve slopes of the wings having aspect ratios of 8 and 6 have been corrected for deflection under load by combining static-loading test results with aerodynamic strip theory and the corrections are presented in table II. The corrections for the wing having an aspect ratio of 4 are negligible and no corrections have been applied. No attempt has been made to correct the aerodynamic-center positions for deflections under load since such corrections are so critically dependent upon both the spanwise and chordwise loadings which are unknown in the transonic range. An estimate based on potential flow indicates the maximum correction to be about 7 percent of the mean aerodynamic chord.

## RESULTS AND DISCUSSION

The force and moment data are presented in figures 6, 7, and 8 and a summary of the aerodynamic characteristics throughout the test Mach number range is shown in figure 9. The slopes summarized in figure 9 have been averaged over the lift-coefficient range from approximately -0.1 to +0.1. The lift-curve slopes presented in figure 9 have been corrected for deflection under load (see section entitled "Apparatus and Tests").

### Lift Characteristics

The variation of the lift-curve slope (fig. 9) with Mach number for all three aspect ratios is characterized by an extremely large loss of lift-curve slope in the Mach number range from about 0.85 to about 1.00 due to shock stall. The reason for the different type of lift-curve-slope variation of the wing having an aspect ratio of 8 below a Mach number of 0.85 is not known. Above a Mach number of 1.00 there is a rapid recovery of lift-curve slope up to a Mach number of about 1.08 as the shock moves to the rear of the airfoil and above this Mach number the lift-curve slope remains essentially constant up to the highest Mach number tested. The slopes presented in figure 9 are for the low-lift range and it will be noted in figures 6 to 8 that in the high-lift range the variation of lift-curve slope with Mach number is considerably less. The experimental lift-curve slopes at a Mach number of 0.70 are compared with the theoretical values determined by the Weissinger method (reference 3) in the following table:

| A | $\frac{\partial C_L}{\partial \alpha}$ (Theoretical) | $\frac{\partial C_L}{\partial \alpha} \times 1.08$ (Theoretical) | $\frac{\partial C_L}{\partial \alpha}$ (Experimental) |
|---|---|---|---|
| 8 | 0.070 | 0.076 | 0.074 |
| 6 | .065 | .070 | .072 |
| 4 | .062 | .067 | .065 |

It will be noted that the Weissinger values are somewhat lower than the experimental values for all three aspect ratios. However a comparison of some Weissinger solutions with some Falkner lifting-surface solutions (references 4 and 5) indicates that for the sweep angle of these wings the Weissinger solutions are probably about 8 percent low. When the Weissinger values are increased by 8 percent, the agreement between the theory and experiment is somewhat better.

Although the accuracy of the bending-moment data, as reflected by the point scatter, is rather poor, it indicates a large inboard shift of lift around a Mach number of 1.00 in the low-lift-coefficient range (see figs. 6 to 8).

## Drag Characteristics

The variation of the drag coefficient at zero lift coefficient with Mach number for the three wings tested is presented in figure 9. At a Mach number of 0.70 all three wings have a drag coefficient of about 0.009. All three wings have a rather gradual drag rise starting in the Mach number range from about 0.90 to 0.95. The drag of the wing having an aspect ratio of 8 levels off to a value of about 0.036, whereas the wings having aspect ratios of 6 and 4 level off to a value of about 0.047. The inferior drag characteristics of the wings having aspect ratios of 6 and 4 might possibly be due to the fact that, as the aspect ratio decreases, the loss of sweep effect over the inboard region of the wing affects a larger percent of the wing. The rather unusual variation of drag coefficient with lift coefficient at a Mach number of 1.00 (see fig. 6 for example) is due to the large decrease in lift-curve slope in the low-lift range at this Mach number.

## Pitching-Moment Characteristics

The variation of the aerodynamic center with Mach number for all three aspect ratios in the lift-coefficient range of ±0.10 is characterized by extremely large forward movements in the Mach number range from about 0.85 to about 1.00 (fig. 9). Above a Mach number of 1.00 there is a rapid rearward movement of the aerodynamic center for all three aspect ratios. The variation of the aerodynamic center with Mach number is, to a large extent, due to variations in the lateral center of pressure with Mach number. Although the variation of the aerodynamic center with Mach number decreased as the aspect ratio was reduced, it is rather large even for the wing having an aspect ratio of 4. In the Mach number range from 0.85 to 1.00 the aerodynamic center in the low-lift range of the wing having an aspect ratio of 4 moved forward approximately 46 percent of the mean aerodynamic chord, whereas the movements for the wings of aspect ratios 6 and 8 were in excess of 100 percent of the mean aerodynamic chord. In the Mach number range from about 0.95 to about 1.05 the variation of pitching-moment coefficient with lift coefficient is very nonlinear (figs. 6 to 8), and above a lift coefficient of about 0.10 the pitching-moment characteristics as a function of Mach number are considerably better especially for the wing having an aspect ratio of 4. Although the changes in taper ratio could have some effect, it is believed that the results presented are due mainly to the change in aspect ratio.

## CONCLUSIONS

The results of an investigation of tapered wings having aspect ratios 4, 6, and 8, quarter-chord lines swept 45°, and NACA $63_1A012$ airfoil sections indicate the following conclusions:

1. The variation of the lift-curve slope in the low-lift range with Mach number for all three aspect ratios was characterized by an extremely large decrease in lift-curve slope in the Mach number range from about 0.85 to about 1.00.

2. All three wings had a rather gradual drag rise starting in the Mach number range from about 0.90 to about 0.95.

3. The variation of the aerodynamic center with Mach number for all three aspect ratios, and especially the wing having an aspect ratio of 8, is characterized in the low-lift range by extremely large forward movements in the Mach number range from about 0.85 to 1.00 followed by a rapid rearward movement. However, above a lift coefficient of about 0.10 the pitching-moment characteristics as a function of Mach number are considerably better especially for the wing having an aspect ratio of 4.

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5. Falkner, V. M.: The Solution of Lifting Plane Problems by Vortex Lattice Theory. Rep. No. 10,895, British A.R.C., Sept. 29, 1947.

TABLE I

NACA $63_1$A012 AIRFOIL ORDINATES

| X (Percent chord) | Y (Percent chord) |
|---|---|
| 0 | 0 |
| .50 | .973 |
| .75 | 1.173 |
| 1.25 | 1.492 |
| 2.50 | 2.078 |
| 5.00 | 2.895 |
| 7.50 | 3.504 |
| 10 | 3.994 |
| 15 | 4.747 |
| 20 | 5.287 |
| 25 | 5.664 |
| 30 | 5.901 |
| 35 | 5.995 |
| 40 | 5.957 |
| 45 | 5.792 |
| 50 | 5.517 |
| 55 | 5.148 |
| 60 | 4.700 |
| 65 | 4.186 |
| 70 | 3.621 |
| 75 | 3.026 |
| 80 | 2.426 |
| 85 | 1.826 |
| 90 | 1.225 |
| 95 | .625 |
| 100 | .025 |
| Leading-edge radius: 1.071<br>Trailing-edge radius: 0.028 | |

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TABLE II

PERCENT-INCREASE CORRECTIONS TO $\frac{\partial C_L}{\partial \alpha}$ FOR DEFLECTION UNDER LOAD

| M | A = 8 | A = 6 |
|---|---|---|
| 0.70 | 9.0 | 3.8 |
| .80 | 9.8 | 5.0 |
| .85 | --- | 5.8 |
| .90 | 12.0 | 5.4 |
| .93 | 11.0 | --- |
| .95 | 9.0 | 3.5 |
| .98 | 8.0 | --- |
| 1.00 | 5.0 | 2.3 |
| 1.05 | 7.8 | 4.0 |
| 1.10 | 9.2 | 4.8 |
| 1.15 | 10.0 | 4.8 |

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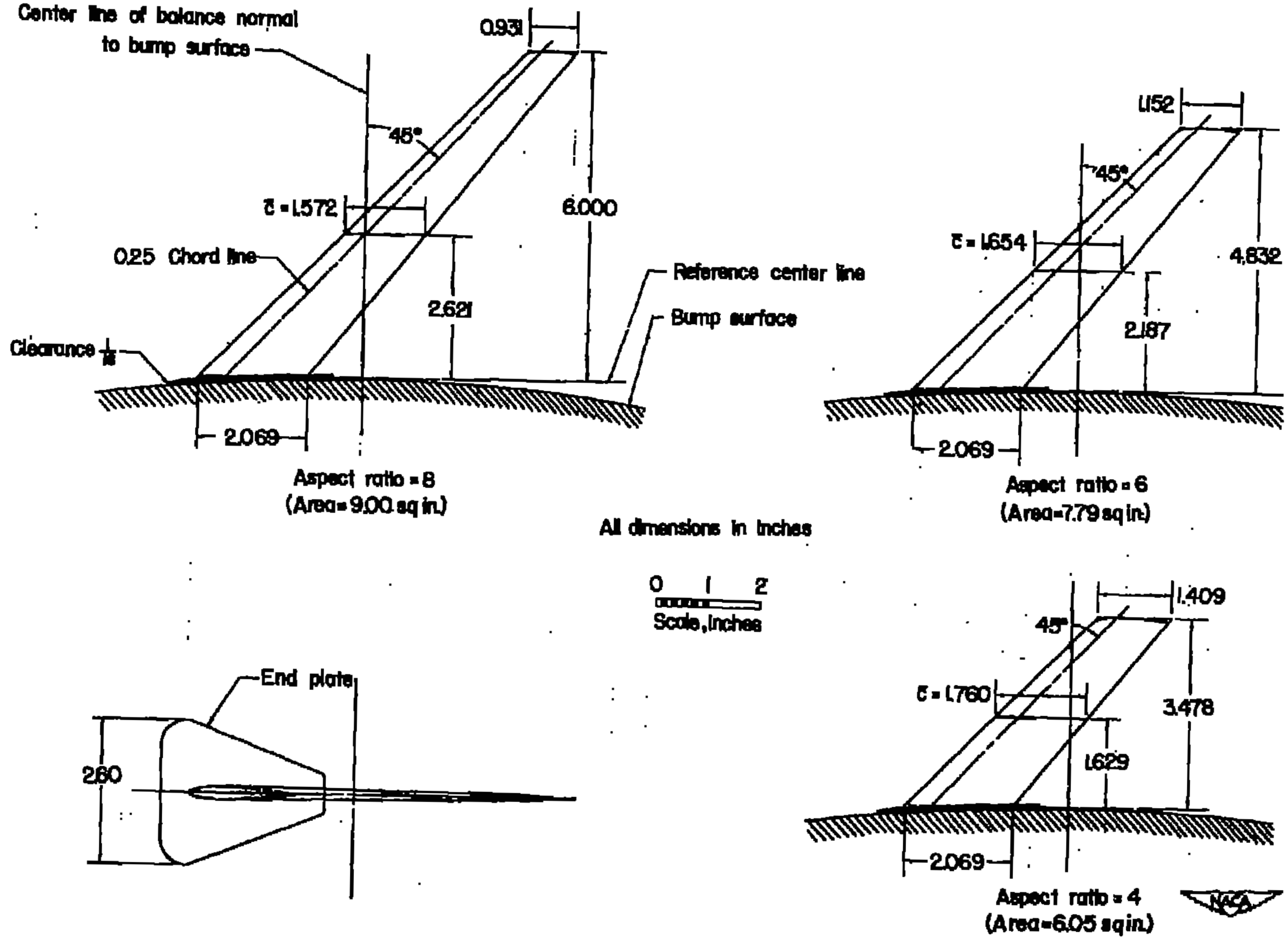


Figure 1.- Plan-form drawings of tapered wings having aspect ratios 4, 6, and 8, quarter-chord lines swept back 45°, and NACA $63_1$A012 airfoil sections.

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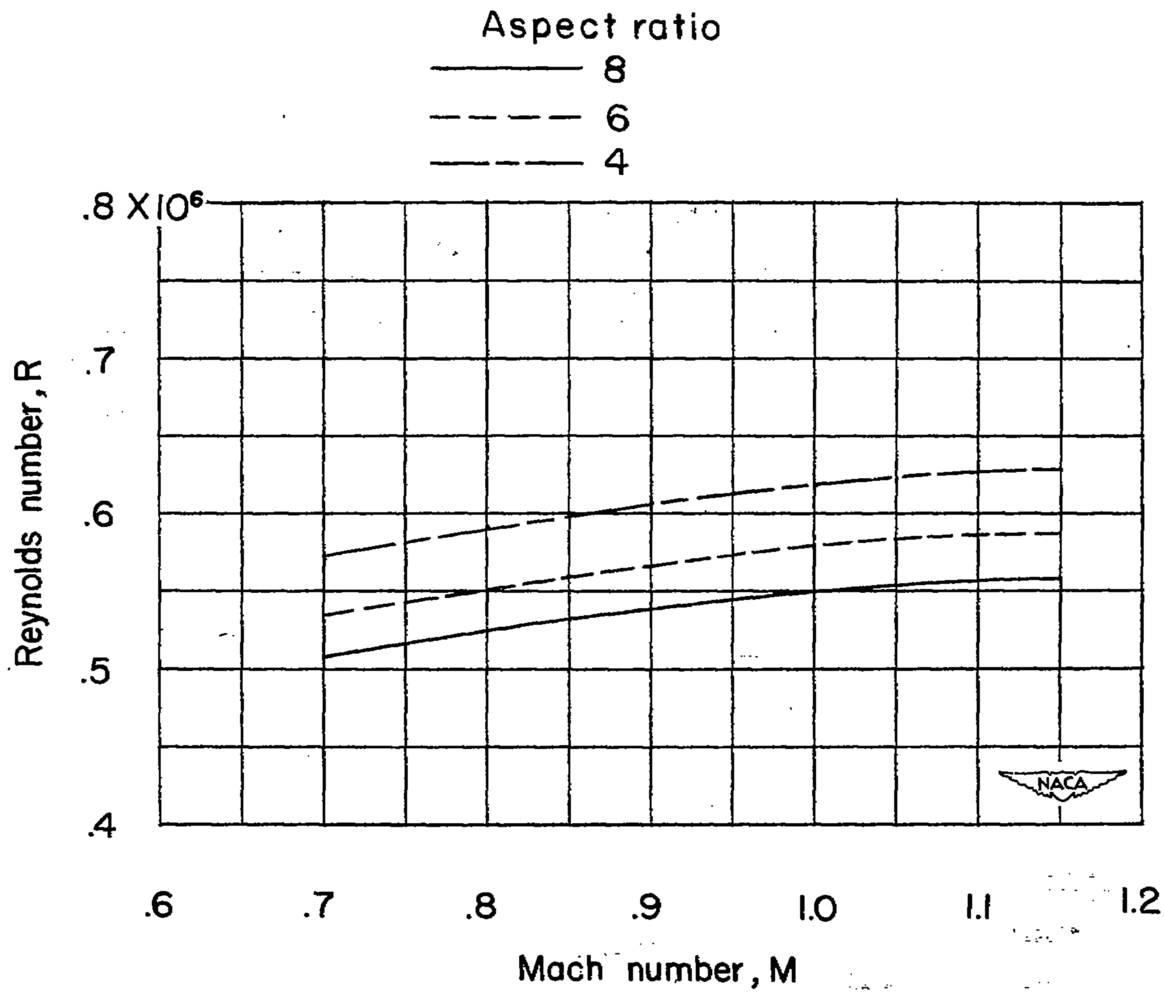


Figure 4.- Variation of test Reynolds number with Mach number for wings with 45° sweepback and NACA $63_1$A012 airfoil section.

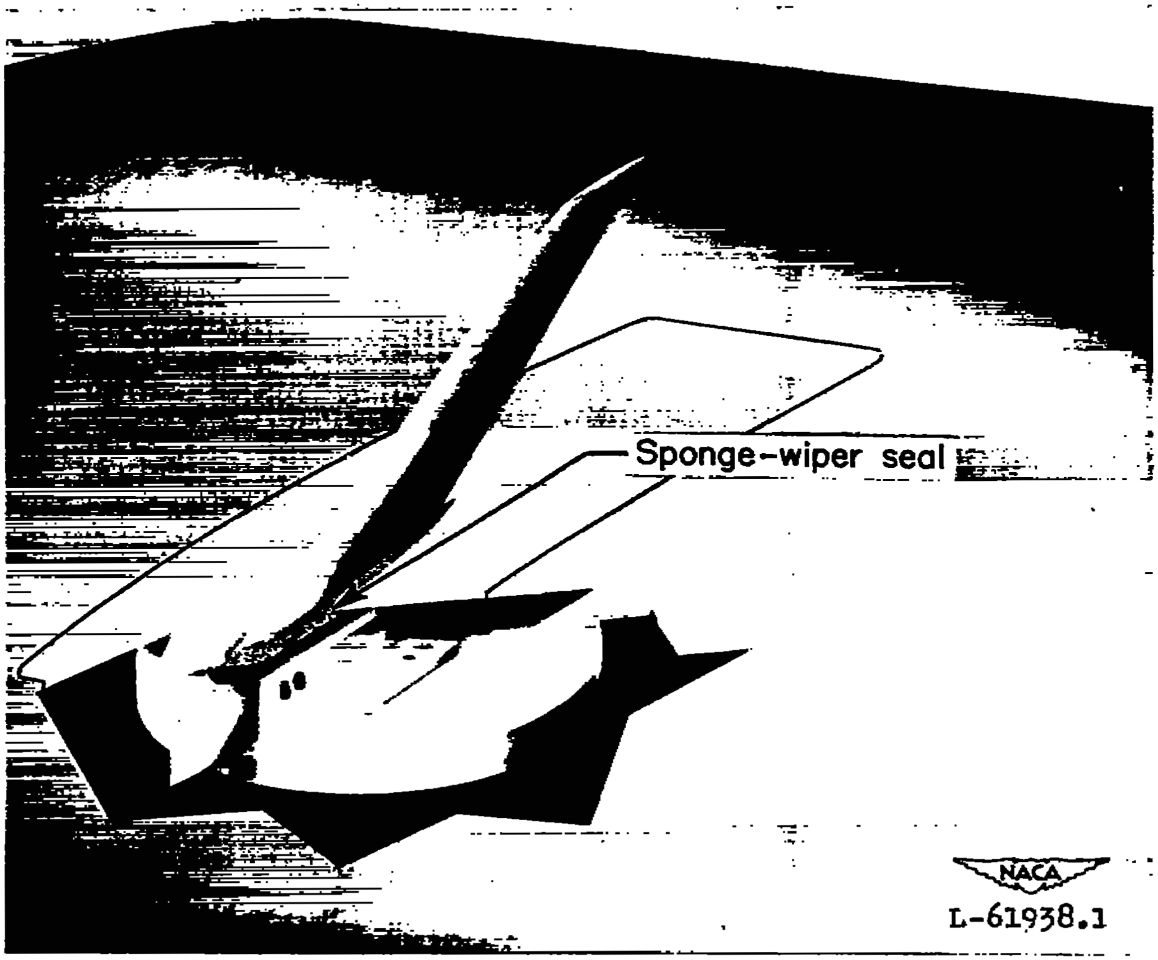


Figure 5.- Pictorial view showing sponge-wiper-seal installation on the model. $\alpha = 0^{\circ}$.

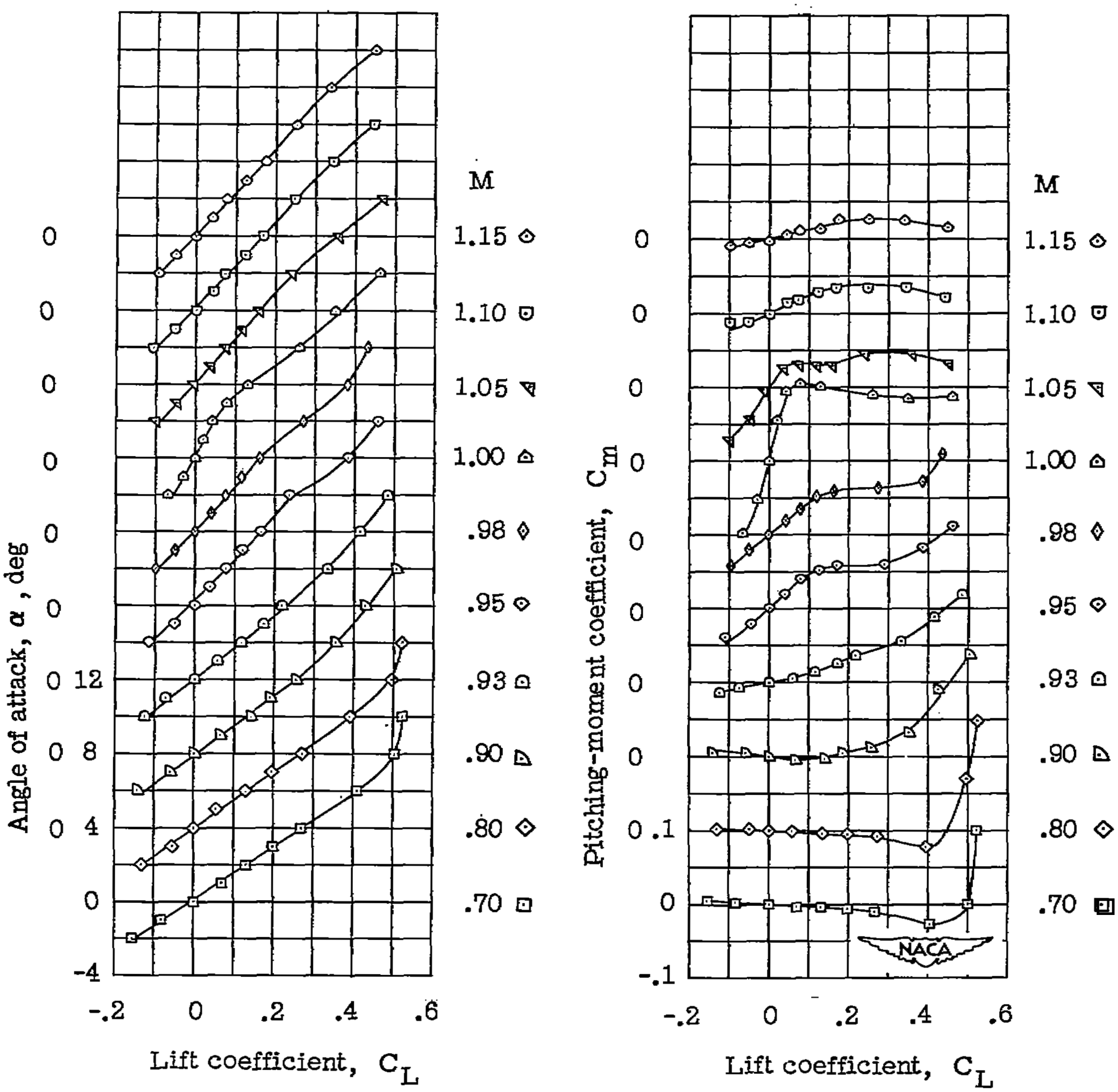


Figure 6.- Aerodynamic characteristics of a 45° sweptback wing with aspect ratio 8, taper ratio 0.45, and NACA $63_1$A012 airfoil.

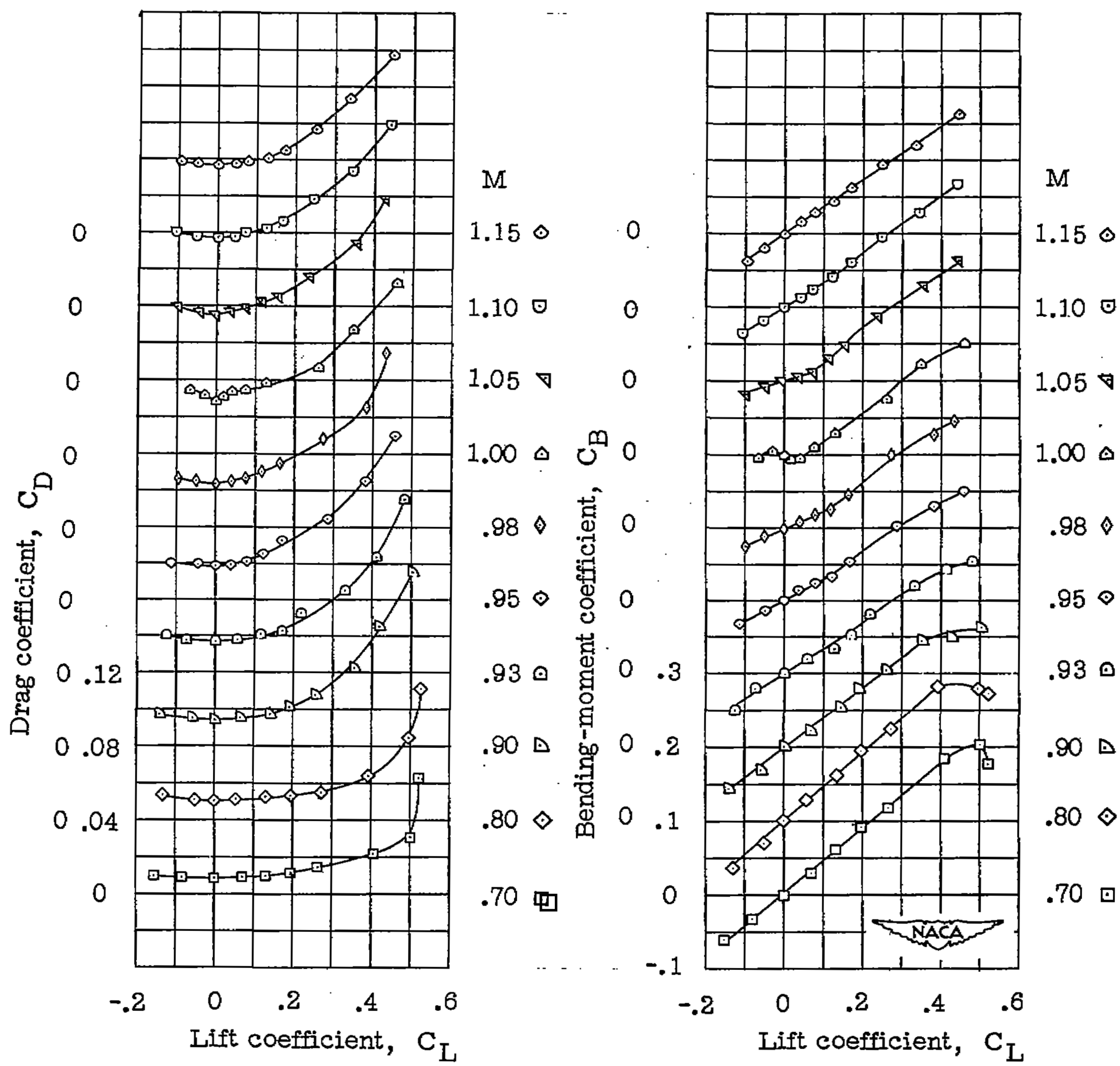


Figure 6.- Concluded.

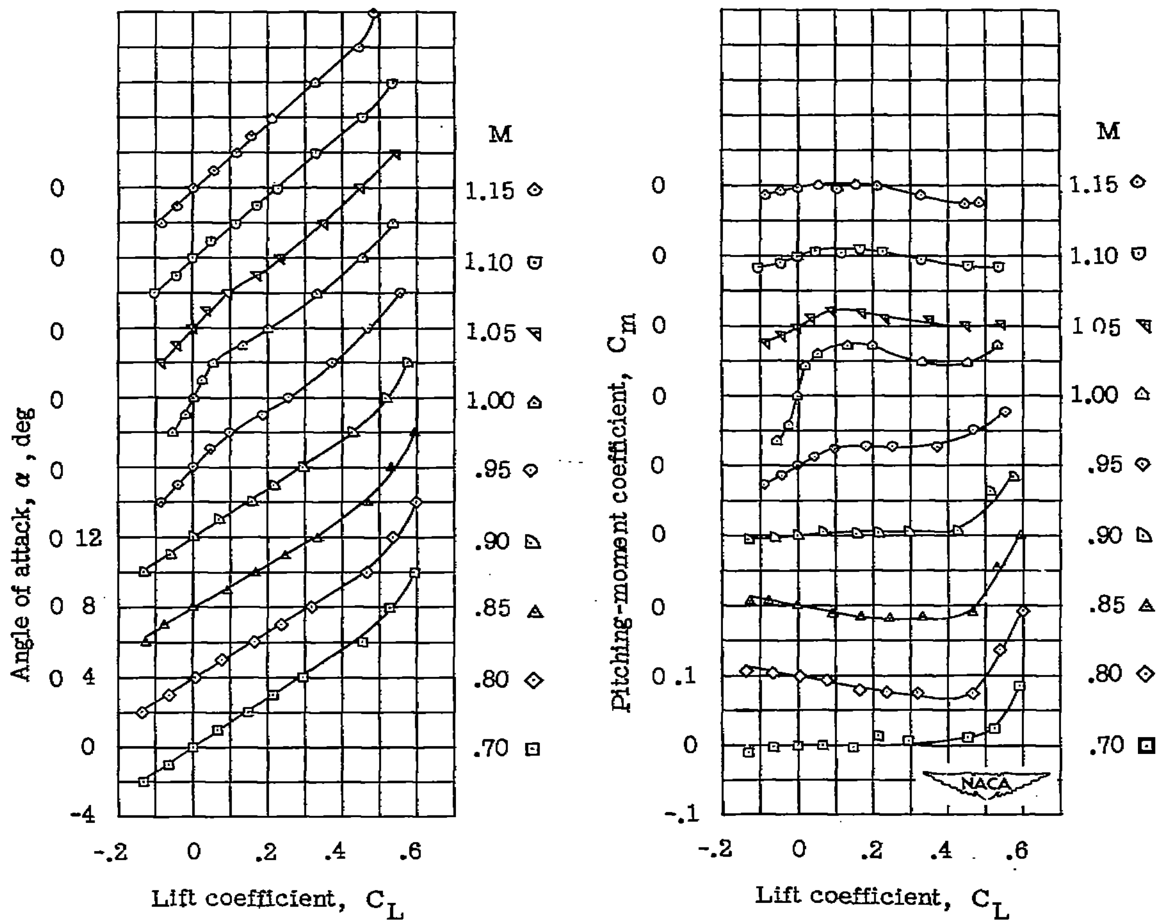


Figure 7.- Aerodynamic characteristics of a 45° sweptback wing with aspect ratio 6, taper ratio 0.56, and NACA $63_1$A012 airfoil.

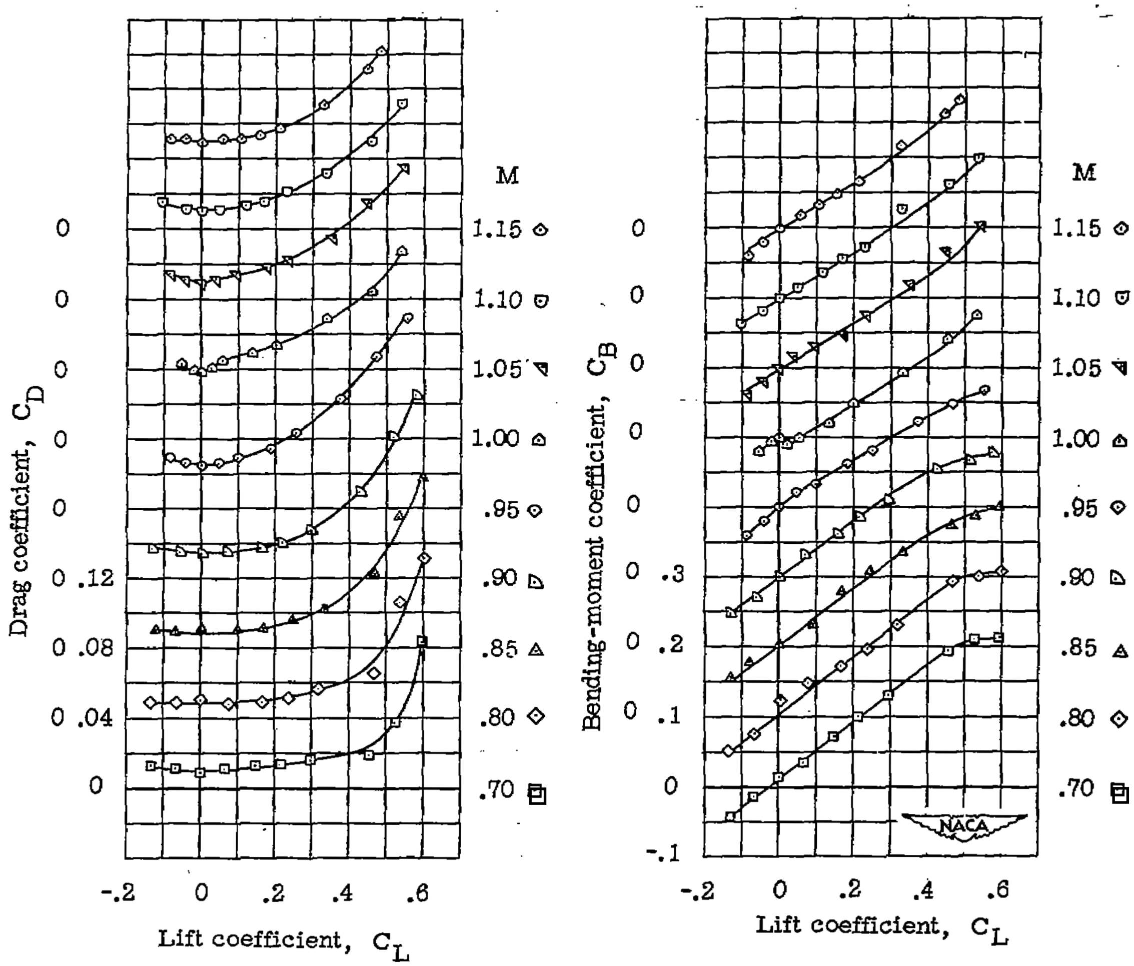


Figure 7.- Concluded.

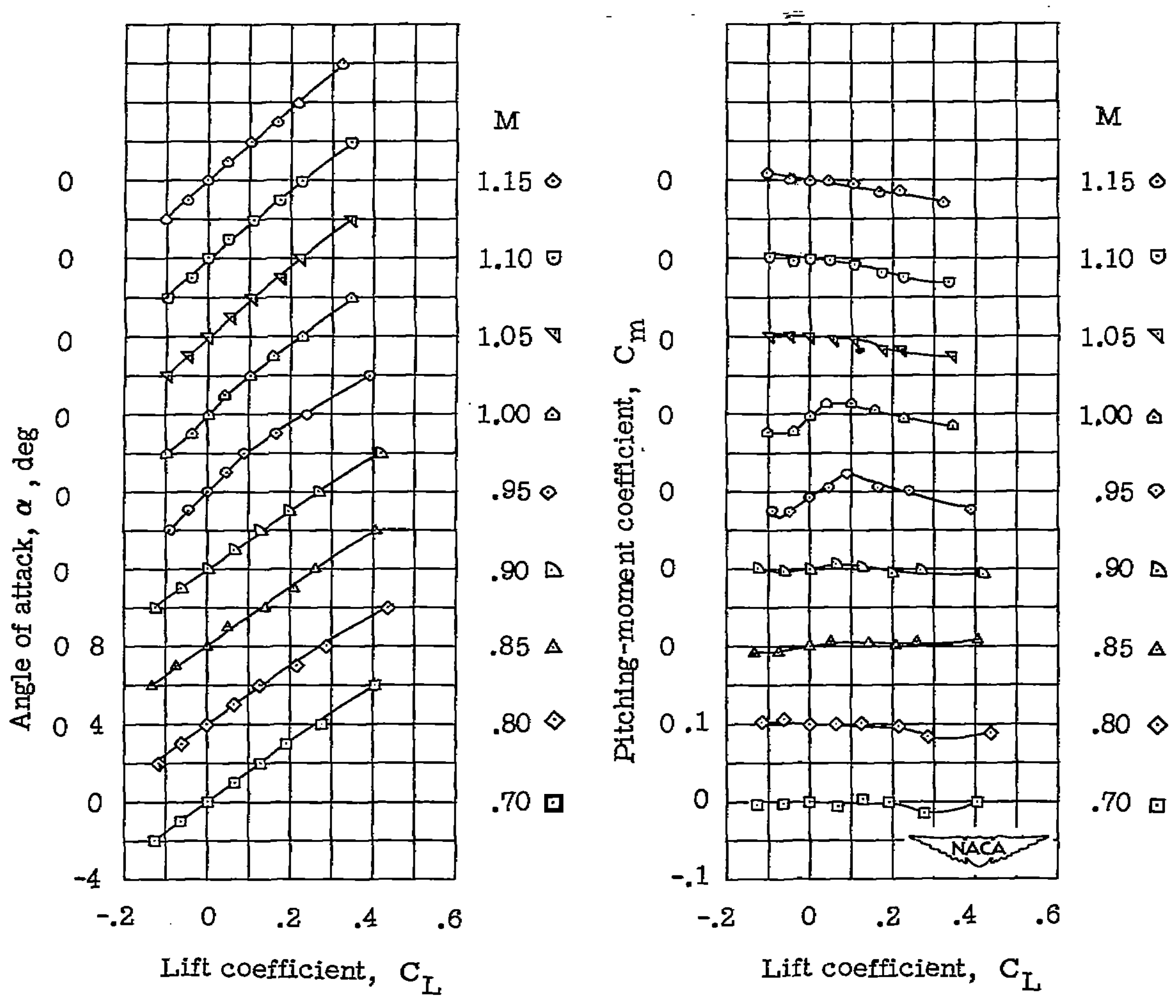


Figure 8.- Aerodynamic characteristics of a 45° sweptback wing with aspect ratio 4, taper ratio 0.68, and NACA $63_1$A012 airfoil.

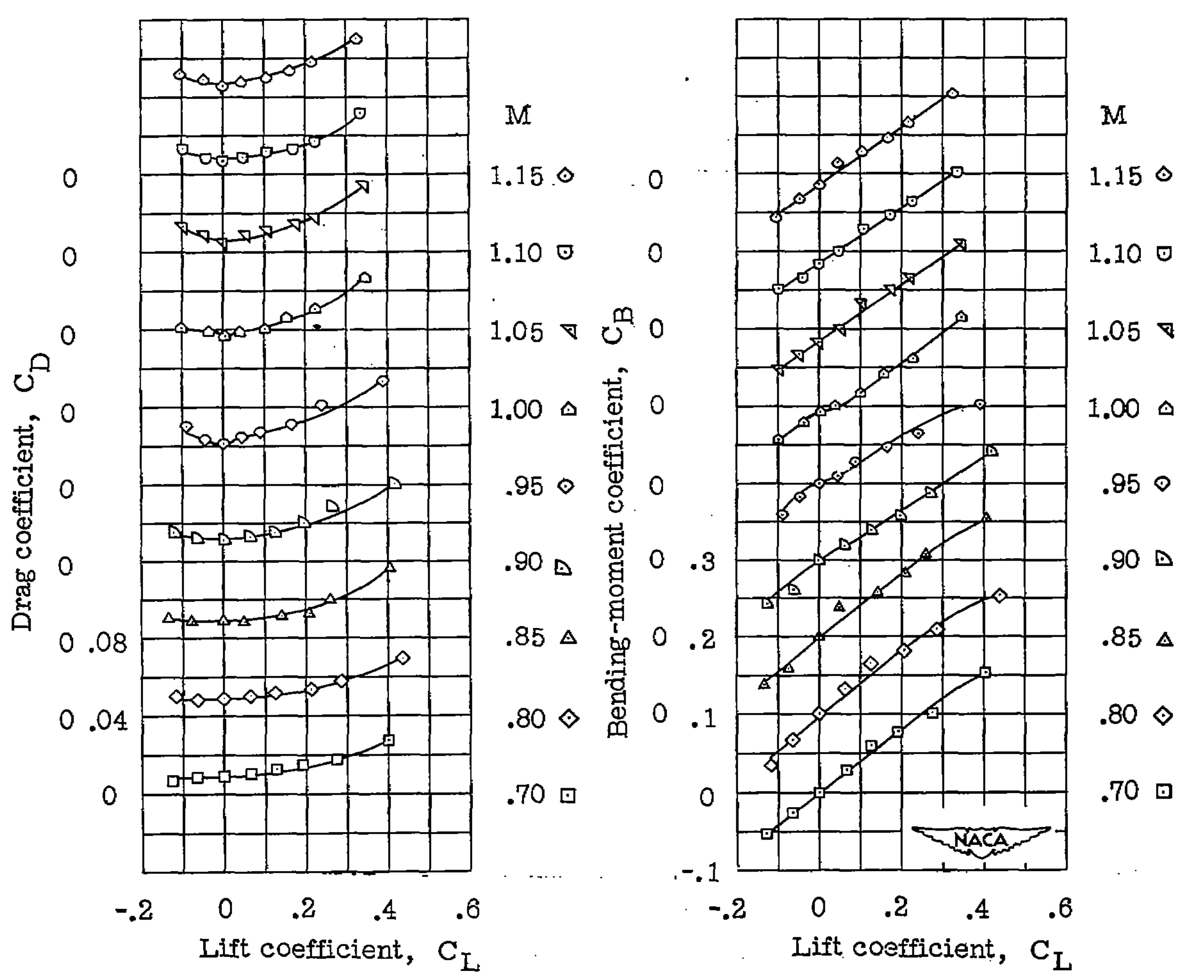


Figure 8.- Concluded.

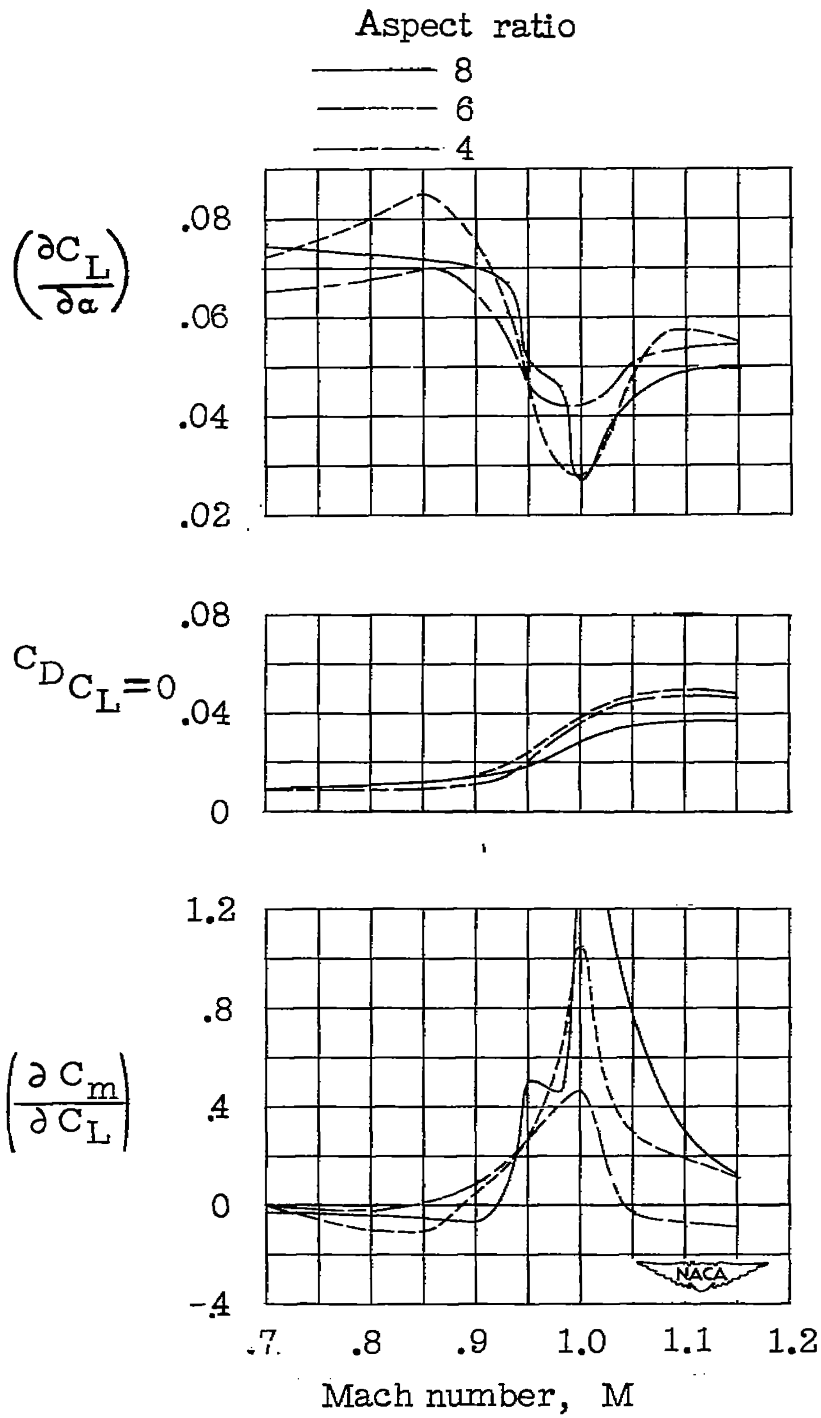


Figure 9.- Lift-curve slopes, drag coefficient at zero lift, and aerodynamic-center location of wings with 45° sweepback and NACA $63_1$A012 airfoil section.



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